Engineering Innovations

Propulsion

Thermal Protection Systems

Materials and Manufacturing

Aerodynamics and Flight Dynamics

Avionics, Navigation, and Instrumentation

Software

Structural Design

Robotics and Automation

Systems Engineering for Life Cycle of Complex Systems

Propulsion

Introduction
Yolanda Harris

Space Shuttle Main Engine
Fred Jue

Chemochromic Hydrogen Leak Detectors
Luke Roberson
Janine Captain
Martha Williams
Mary Whitten

The First Human-Rated Reuseable Solid Rocket Motor
Fred Perkins
Holly Lamb

Orbital Propulsion Systems
Cecil Gibson
Willard Castner
Robert Cort
Samuel Jones

Pioneering Inspection Tool
Mike Lingbloom

Propulsion Systems and Hazardous Gas Detection
Bill Helms
David Collins
Ozzie Fish
Richard Mizell

The launch of the Space Shuttle was probably the most visible event of the entire mission cycle. The image of the Main Propulsion System—the Space Shuttle Main Engine and the Solid Rocket Boosters (SRBs)—powering the Orbiter into space captured the attention and the imagination of people around the globe. Even by 2010 standards, these main engines' performance was unsurpassed compared to any other engines. They were a quantum leap from previous rocket engines. The main engines were the most reliable and extensively tested rocket engine before and during the shuttle era.

The shuttle's SRBs were the largest ever used, the first reusable rocket, and the only solid fuel certified for human spaceflight. This technology, engineering, and manufacturing may remain unsurpassed for decades to come.

But the shuttle's propulsion capabilities also encompassed the Orbiter's equally important array of rockets—the Orbital Maneuvering System and the Reaction Control System—which were used to fine-tune orbits and perform the delicate adjustments needed to dock the Orbiter with the International Space Station. The design and maintenance of the first reusable space vehicle—the Orbiter—presented a unique set of challenges. In fact, the Space Shuttle Program developed the world's most extensive materials database for propulsion. In all, the shuttle's propulsion systems achieved unprecedented engineering milestones and launched a 30-year era of American space exploration.

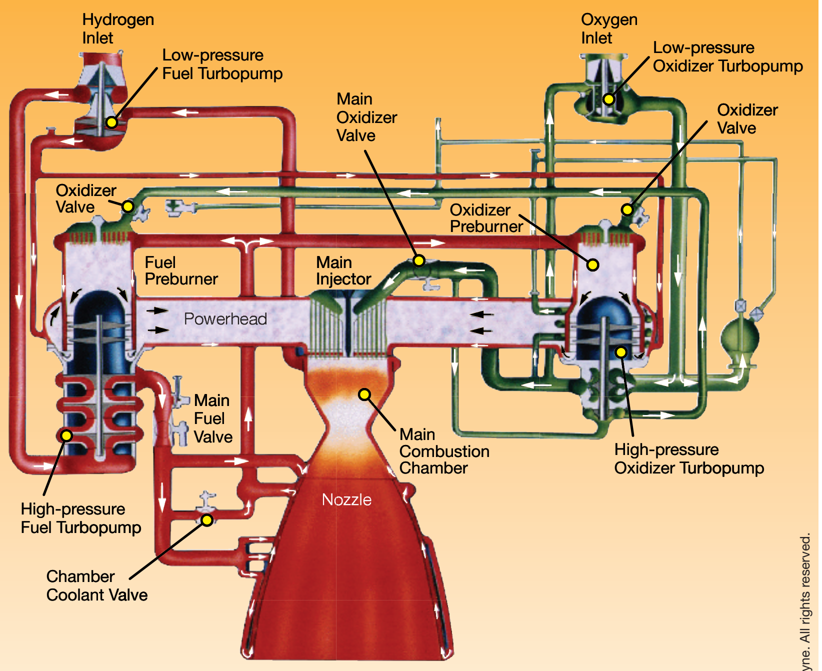
Space Shuttle Main Engine

NASA faced a unique challenge at the beginning of the Space Shuttle Program: to design and fly a human-rated reusable liquid propulsion rocket engine to launch the shuttle. It was the first and only liquid-fueled rocket engine to be reused from one mission to the next during the shuttle era. The improvement of the Space Shuttle Main Engine (SSME) was a continuous undertaking, with the objectives being to increase safety, reliability, and operational margins; reduce maintenance; and improve the life of the engine's high-pressure turbopumps.

The reusable SSME was a staged combustion cycle engine. Using a mixture of liquid oxygen and liquid hydrogen, the main engine could attain a maximum thrust level (in vacuum) of 232,375 kg (512,300 pounds), which is equivalent to greater than 12,000,000 horsepower (hp). The engine also featured high-performance fuel and oxidizer turbopumps that developed 69,000 hp and 25,000 hp, respectively. Ultra-high-pressure operation of the pumps and combustion chamber allowed expansion of hot gases through the exhaust nozzle to achieve efficiencies never previously attained in a rocket engine.

Requirements established for Space Shuttle design and development began in the mid 1960s. These requirements called for a two-stage-to-orbit vehicle configuration with liquid oxygen (oxidizer) and liquid hydrogen (fuel) for the Orbiter's main engines. By 1969, NASA awarded advanced engine studies to three contractor firms to further define designs necessary to meet the leap in performance demanded by the new Space Transportation System (STS).

Space Shuttle Main Engine Propellant Flow



The Space Shuttle Main Engine used a two-stage combustion process. Liquid hydrogen and liquid oxygen were pumped from the External Tank and burned in two preburners. The hot gases from the preburners drove two high-pressure turbopumps—one for liquid hydrogen (fuel) and one for liquid oxygen (oxidizer).

In 1971, the Rocketdyne division of Rockwell International was awarded a contract to design, develop, and produce the main engine.

The main engine would be the first production-staged combustion cycle engine for the United States. (The Soviet Union had previously demonstrated the viability of staged combustion cycle in the Proton vehicle in 1965.) The staged combustion cycle yielded high efficiency in a technologically advanced and complex engine that operated at pressures beyond known experience.

The design team chose a dual-preburner powerhead configuration to provide precise mixture ratio and throttling control. A low- and high-pressure turbopump, placed in series for each of the liquid hydrogen and liquid oxygen loops, generated high pressures across a wide range of power levels.

A weight target of 2,857 kg (6,300 pounds) and tight Orbiter ascent envelope requirements yielded a compact design capable of generating a nominal chamber pressure of 211 kg/cm^2 (3,000 pounds/in^2)—about four times that of the Apollo/Saturn J-2 engine.

Michael Coats
Pilot on STS-41D (1984).
Commander on STS-29 (1989) and STS-39 (1991).

A Balky Hydrogen Valve Halts Discovery Liftoff

"I had the privilege of being the pilot on the maiden flight of the Orbiter Discovery, a hugely successful mission. We deployed three large communications satellites and tested the dynamic response characteristics of an extendable solar array wing, which was a precursor to the much-larger solar array wings on the International Space Station.

"But the first launch attempt did not go quite as we expected. Our pulses were racing as the three main engines sequentially began to roar to life, but as we rocked forward on the launch pad it suddenly got deathly quiet and all motion stopped abruptly. With the seagulls screaming in protest outside our windows, it dawned on us we weren't going into space that day. The first comment came from Mission Specialist Steve Hawley, who broke the stunned silence by calmly saying 'I thought we'd be a lot higher at MECO (main engine cutoff).' So we soon started cracking lousy jokes while waiting for the ground crew to return to the pad and open the hatch. The joking was short-lived when we realized there was a residual fire coming up the left side of the Orbiter, fed from the same balky hydrogen valve that had caused the abort. The Launch Control Center team was quick to identify the problem and initiated the water deluge system designed for just such a contingency. We had to exit the pad elevator through a virtual wall of water. We wore thin, blue cotton flight suits back then and were soaked to the bone as we entered the air-conditioned astronaut van for the ride back to crew quarters. Our drenched crew shivered and huddled together as we watched the Discovery recede through the rear window of the van, and as Mike Mullane wryly observed, 'This isn't exactly what I expected spaceflight to be like.' The entire crew, including Commander Henry Hartsfield, the other Mission Specialists Mike Mullane and Judy Resnik, and Payload Specialist Charlie Walker, contributed to an easy camaraderie that made the long hours of training for the mission truly enjoyable."

For the first time in a boost-to-orbit rocket engine application, an on-board digital main engine controller continuously monitored and controlled all engine functions. The controller initiated and monitored engine parameters and adjusted control valves to maintain the performance parameters required by the mission. When detecting a malfunction, it also commanded the engine into a safe lockup mode or engine shutdown.

Design Challenges

Emphasis on fatigue capability, strength, ease of assembly and disassembly, maintainability, and materials compatibility were all major considerations in achieving a fully reusable design.

Specialized materials needed to be incorporated into the design to meet the severe operating environments. NASA successfully adapted advanced alloys, including cast titanium, Inconel® 718 (a high-strength, nickel-based superalloy used in the main combustion chamber support jacket and powerhead), and NARloy-Z (a high-conductivity, copper-based alloy used as the liner in the main combustion chamber). NASA also oversaw the development of single-crystal turbine blades for the high-pressure turbopumps. This innovation essentially eliminated the grain boundary separation failure mechanism (blade cracking) that had limited the service life of the pumps. Nonmetallic materials such as Kel-F® (a plastic used in turbopump seals), Armalon® fabric (turbopump bearing cage material), and P5N carbon-graphite seal material were also incorporated into the design.

Material sensitivity to oxygen environment was a major concern for compatibility due to reaction and ignition under the high pressures. Mechanical impact testing had vastly expanded in the 1970s to accommodate the shuttle engine's varied operating conditions. This led to a new class of liquid oxygen reaction testing up to 703 kg/cm^2 (10,000 pounds/in^2).

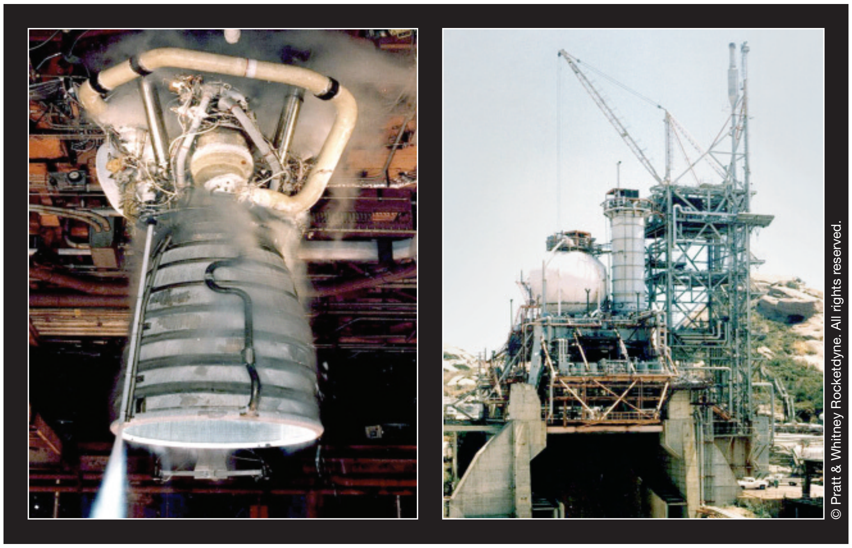
Engineers also needed to understand long-term reaction to hydrogen effects to achieve full reusability. Thus, a whole field of materials testing evolved to evaluate the behavior of hydrogen charging on all affected materials.

NASA developed new tools to accomplish design advancements. Engineering design tools advanced along with the digital age as analysis migrated from the mainframe platform to workstations and desktop personal computers. Fracture mechanics and fracture control became critical tools for understanding the characteristics of crack propagation to ensure design reusability. As the analytical tools and processor power improved over the decades, cycle time for engineering analysis such as finite element models, computer-aided design and manufacturing, and computational fluid dynamics dropped from days to minutes. Real-time engine performance analyses were conducted during ground tests and flights at the end of the shuttle era.

Development and Certification

The shuttle propulsion system was the most critical system during ascent; therefore, a high level of testing was needed prior to first flight to demonstrate engine maturity. Component-level testing of the preburners and thrust chamber began in 1974 at Rocketdyne's Santa Susana Field Laboratory in Southern California.

The first engine-level test of the main engine—the Integrated Subsystem Test Bed—occurred in 1975 at the NASA National Space Technology Laboratory (now Stennis Space Center) in Mississippi and relied on facility controls, as the main engine controller was not yet available.



A 1970s-era Space Shuttle Main Engine undergoes testing at Rocketdyne's Santa Susana Field Laboratory near Los Angeles, California.

NASA and Rocketdyne pursued an aggressive test schedule at their respective facilities. Stennis Space Center with three test stands and Rocketdyne with one test stand completed 152 engine tests in 1980 alone—a record that has not been exceeded since. This ramp-up to 100,000 seconds represented a team effort of personnel and facilities to overachieve a stated development goal of 65,000 seconds set by then-Administrator John Yardley as the maturity level deemed flightworthy. NASA verified operation at altitude conditions and also demonstrated the rigors of sea-level performance and engine gimballing for thrust vector control. The Rocketdyne laboratory supplemented sea-level testing as well as deep throttling by using a low 33:1 expansion ratio nozzle. This testing was crucial in identifying shortcomings related to the initial design of the high-pressure turbopumps, powerhead, valves, and nozzles.

Extensive margin testing beyond the normal flight envelope—including high-power, extended-duration tests and near-depleted inlet propellant conditions to simulate the effects of microgravity—provided further confidence in the design. Engineers subjected key components to a full series of design verification tests, some with intentional hardware defects, to validate safety margins should the components develop undetected flaws during operation.

NASA and Rocketdyne also performed system testing to replicate the three engine cluster interactions with the Orbiter. The Main Propulsion Test Article consisted of an Orbiter aft fuselage, complete with full thrust structure, main propulsion electrical and system plumbing, External Tank, and three main engines. To validate that the Main Propulsion System was ready for launch, engineers completed 18 tests at the National Space Technology Laboratory by 1981.

The completion of the main engine preliminary flight certification in March 1981 marked a major milestone in clearing the initial flights at 100% rated power level.

Design Evolutions

A major requirement in engine design was the ability to operate at various power levels. The original engine life requirement was 100 nominal missions and 27,000 seconds (7.5 hours) of engine life. Nominal thrust, designated as rated power level, was 213,189 kg (470,000 pounds) in vacuum. The life requirement included six exposures at the emergency power level of 232,375 kg (512,300 pounds), which was designated 109% of rated power level. To maximize the number of missions possible at emergency power level, an assessment of the engine capability resulted in reducing the number of nominal missions per engine to 55 missions at 109%. Emergency power level was subsequently renamed full power level.

Ongoing ascent trajectory analysis determined 65% of rated power level to be sufficient to power the vehicle through its period of maximum aerodynamic pressure during ascent. Minimum power level was later refined upward to 67%.

On April 12, 1981, Space Shuttle Columbia lifted off Launch Pad 39A from Kennedy Space Center in Florida on its maiden voyage. The first flight configuration engines were aptly named the First Manned Orbital Flight SSMEs. These engines were flown during the initial five shuttle development missions at 100% rated power level thrust. Work done to prepare for the next flight validated the ability to perform routine engine maintenance without removing them from the Orbiter.

The successful flight of STS-1 initiated the development of a full-power (109% rated power level) engine. The higher thrust capability was needed to support an envisioned multitude of NASA, commercial, and Department of Defense payloads, especially if the shuttle was launched from the West Coast. By 1983, however, test failures demonstrated the basic engine lacked margin to continuously operate at 109% thrust, and full-power-level development was halted. Other engine improvements were implemented into what was called the Phase II engine. During this period, the engine program was restructured into two programs—flight and development.

Post-Challenger Return to Flight

The 1986 Challenger accident provoked fundamental changes to the shuttle, including an improved main engine called Phase II. This included changes to the high-pressure turbopumps and main combustion chamber, avionics, valves, and high-pressure fuel duct insulation. An additional 90,241 seconds of engine testing accrued, including recertification to 104% rated power level.

The new Phase II engine continued to be the workhorse configuration for shuttle launches up to the late 1990s while additional improvements envisioned during the 1980s were undergoing development and flight certification for later incorporation. NASA targeted five major components for advanced development to further enhance safety and reliability, lower recurring costs, and increase performance capability. These components included the powerhead, heat exchanger, main combustion chamber, and high-pressure oxidizer and fuel turbopumps.

These major changes would later be divided into two "Block" configuration upgrades, with Rocketdyne tasked to improve the powerhead, heat exchanger, and main combustion chamber while Pratt & Whitney was selected to design, develop, and produce the improved high-pressure turbopumps.

Pratt & Whitney Company of United Technologies began the effort in 1986 to provide alternate high-pressure turbopumps as direct line replaceable units for the main engines. Pratt & Whitney used staged combustion experience from its development of the XLR-129 engine for the US Air Force and cryogenic hydrogen experience from the RL-10 (an upper-stage engine used by NASA, the military, and commercial enterprises) along with SSME lessons learned to design the new pumps. The redesign of the components eliminated critical failure modes and increased safety margins.

Next Generation

The Block I configuration became the successor to the Phase II engine. A new Pratt & Whitney high-pressure oxygen turbopump, an improved two-duct engine powerhead, and a single-tube heat exchanger were introduced that collectively used new design and production processes to eliminate failure causes. Also it increased the inherent reliability and operating margin and reduced production cycle time and costs. This Block I engine first flew on STS-70 (1995).

The powerhead redesign was less risky and was chosen to proceed ahead of the main combustion chamber.

The Technology Test Bed Space Shuttle Main Engine test program was conducted at Marshall Space Flight Center, Alabama, between September 1988 and May 1996. The program demonstrated the ability of the main engine to accommodate a wide variation in safe operating ranges.

The two-duct powerhead eliminated 74 welds and had 52 fewer parts. This improved design led to production simplification and a 40% cost reduction compared to the previous three-duct configuration. The two-duct configuration provided an improvement to the hot gas flow field distribution and reductions in dynamic pressures. The improved heat exchanger eliminated all inter-propellant welds, and its wall thickness was increased by 25% for added margin against penetration by unexpected foreign debris impact.

The new high-pressure oxygen turbopump eliminated 293 welds, added improved suction performance, and introduced a stiff single-piece disk/shaft configuration and thin-cast turbine blades. The oxygen turbopump incorporated silicon nitride (ceramic) ball bearings in a rocket engine application and could be serviced without removal from the engine. Initial component-level testing occurred at the Pratt & Whitney West Palm Beach, Florida, testing facilities. Testing then graduated to the engine level at Stennis Space Center as well as at Marshall Space Flight Center's (MSFC's) Technology Test Bed test configuration.

The large-throat main combustion chamber began prototype testing at Rocketdyne in 1988. But it was not until 1992, after a series of combustion stability tests at the MSFC Technology Test Bed facility, that concerns regarding combustion stability were put to rest. The next improved engine—Block II—incorporated the new high-pressure fuel turbopump, modified low-pressure turbopumps, software operability enhancements, and previous Block I upgrades. These upgrades were needed to support International Space Station (ISS) launches with their heavy payloads beginning in 1998.

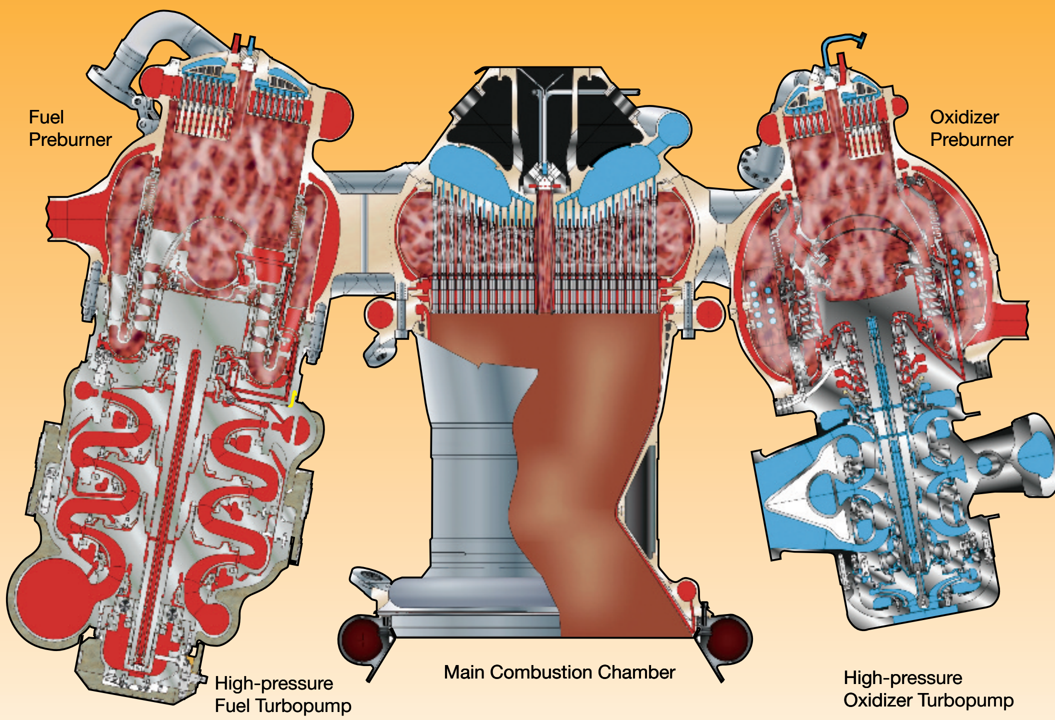
As Block II development testing progressed, the engineering accomplishments on the large-throat main combustion chamber matured more rapidly than the high-pressure fuel turbopump.

By February 1997, NASA had decided to go forward with an interim configuration called the Block IIA. Using the existing Phase II high-pressure fuel pump, this configuration would allow early implementation of the large-throat main combustion chamber to support ISS launches. The large-throat main combustion chamber was simpler and producible. The new chamber lowered the engine's operating pressures and temperatures while increasing the engine's operational safety margin. Changes to the low-pressure turbopumps to operate in this derated environment, along with further avionics improvements, were flown in 1998 on STS-89.

The large-throat main combustion chamber became one of the most significant safety improvements for the main engine by effectively reducing operating pressures and temperatures up to 10% for all subsystems. This design also incorporated improved cooling capability for longer life and used high-strength castings, thus eliminating 50 welds.

By the time the first Block IIA flew on STS-89 in January 1998, the large-throat main combustion chamber design had accumulated in excess of 100,000 seconds of testing time. By late 1999, the Block II high-pressure fuel turbopump had progressed into certification testing. The design philosophy mirrored those proven successful in the high-pressure oxidizer turbopump and included the elimination of 387 welds

The Improved Space Shuttle Main Engine Powerhead Component Arrangement for Block II Engines



The Block II engine combined a new high-pressure fuel turbopump with the previously flown redesigned high-pressure oxygen turbopump. Risk analysis showed that the Block II engine was twice as safe as the 1990s-era engine. Beginning with STS-110 in April 2002, all shuttle flights were powered by the improved Space Shuttle Main Engine.

and incorporation of a stiff single-piece disk/shaft, thin-cast turbine blades, and a cast pump inlet that improved the suction performance and robustness against pressure surges. As with the high-pressure oxidizer turbopump, the high-pressure fuel turbopump turbine inlet did not require off-engine inspections, which contributed significantly to improving engine turnaround time. The high-pressure fuel turbopump also demonstrated that a turbine blade failure would result in a contained, safe engine shutdown. By introducing the added operational margin of the large-throat main combustion chamber with the new turbopumps, quantitative risk analysis projected that the Block II engine was twice as safe as the Phase II engine.

The first two single-engine flights of Block II occurred on STS-104 and STS-108 in July 2001 and December 2001, respectively, followed by the first three-engine cluster flight on STS-110 in April 2002. The high-pressure fuel turbopump had accumulated 150,843 seconds of engine test maturity at the time of the first flight.

The Block II engine also incorporated the advanced health management system on STS-117 in 2007. This on-board system could detect and mitigate anomalous high-pressure turbopump vibration behavior, and the system further improved engine ascent safety by an additional 23%.

Summary

Another major SSME milestone took place in 2004 when the main engine passed 1,000,000 seconds in test and operating time. This unprecedented level of engine maturity over the preceding 3 decades established the main engine as one of the world's most reliable rocket engines, with a 100% flight safety record and a demonstrated reliability exceeding 0.9996 in over 1,000,000 seconds of hot-fire experience.

Chemochromic Hydrogen Leak Detectors

The Chemochromic Point Detector for sensing hydrogen gas leakage is useful in any application in which it is important to know the presence and location of a hydrogen gas leak.

This technology uses a chemochromic pigment and polymer that can be molded or spun into a rigid or pliable shape useable in variable-temperature environments including atmospheres of inert gas, hydrogen gas, or mixtures of gases. A change in the color of detector material reveals the location of a leak. Benefits of this technology include: temperature stability, from -75°C to 100°C (-103°F to 212°F); use in cryogenic applications; ease of application and removal; lack of a power requirement; quick response time; visual or electronic leak detection; nonhazardous qualities, thus environmentally friendly; remote monitoring capability; and a long shelf life. This technology is also durable and inexpensive.

The detector can be fabricated into two types of sensors—reversible and irreversible. Both versions immediately notify the operator of the presence of low levels of hydrogen; however, the reversible version does not require replacement after exposure. Both versions were incorporated into numerous polymeric materials for specific applications including: extruded tapes for wrapping around valves and joints suspected of leaking; injection-molded parts for seals, O-rings, pipe fittings, or plastic piping material; melt-spun fibers for clothing applications; and paint for direct application to ground support equipment. The versatility of the sensor for several different applications provides the operator with a specific-use safety notification while working under hazardous operations.

Hydrogen-sensing tape applied to the Orbiter midbody umbilical unit during fuel cell loading for STS-118 through STS-123 at Kennedy Space Center, Florida.

Hydrogen-sensing tape application at liquid hydrogen cross-country vent line flanges on the pad slope.

The First Human-Rated Reusable Solid Rocket Motor

The Space Shuttle reusable solid rocket motors were the largest solid rockets ever used, the first reusable solid rockets, and the only solids ever certified for crewed spaceflight. The closest solid-fueled rival—the Titan IV Solid Rocket Motor Upgrade—was known for boosting heavy payloads for the US Air Force and National Reconnaissance Organization. The motors were additionally known for launching the 5,586-kg (12,220-pound) Cassini mission on its 7-year voyage to Saturn. By contrast, the Titan booster was 76 cm (30 in.) smaller in diameter and 4.2 m (14 ft) shorter in length, and held only two-thirds of the amount of propellant.

In a class of its own, the Reusable Solid Rocket Motor Program was characterized from its inception by four distinguishing traits: hardware reusability, postflight recovery and analysis, a robust ground-test program, and a culture of continual improvement via process control.

The challenge NASA faced in developing the first human-rated solid rocket motor was to engineer a pair of solid-fueled rocket motors capable of meeting the rigorous reliability requirements associated with human spaceflight. The rocket motors would have to be powerful enough to boost the shuttle system into orbit. The motors would also need to be robust enough to meet stringent reliability requirements and survive the additional rigors of re-entry into Earth's atmosphere and subsequent splashdown, all while being reusable. The prime contractor—Morton Thiokol, Utah—completed its

The two shuttle reusable solid rocket motors, which stood more than 38 m (126 ft) tall, harnessed 29.4 meganewtons (6.6 million pounds) of thrust. The twin solid-fueled rockets provided 80% of the thrust needed to achieve liftoff.

first full-scale demonstration test within 3 years.

NASA learned a poignant lesson in the value of spent booster recovery and inspection with the Challenger tragedy in January 1986. The postflight condition of the hardware provided valuable information on the health of the design and triggered a redesign effort that surpassed, in magnitude and complexity, the original development program.

For the substantial redesign that occurred between 1986 and 1988, engineers incorporated lessons learned from the first 25 shuttle flight booster sets. More than 100 tests, including five full-scale ground tests, were conducted to demonstrate the strength of the new design. Flaws were deliberately manufactured into the final test motor to check redundant systems.

The redesigned motors flew for the first time in September 1988 and performed flawlessly.

A Proven Design

To construct the reusable solid rocket motor, four cylindrical steel segments—insulated and loaded with a high-performance solid propellant—were joined together to form what was essentially a huge pressure vessel and combustion chamber. The segmented design provided maximum flexibility in motor fabrication, transportation, and handling. Each segment measured 3.7 m (12 ft) in diameter and was forged from D6AC steel measuring approximately 1.27 cm (0.5 in.) in thickness.

Case integrity and strength were maintained during flight by insulating the case interior. The insulating liner was a fiber-filled elastomeric (rubber-like) material applied to the interior of the steel cylinders. A carefully formulated tacky rubber bonding layer—or "liner"—was applied to the rubber insulator surface to facilitate a strong bond with the propellant.

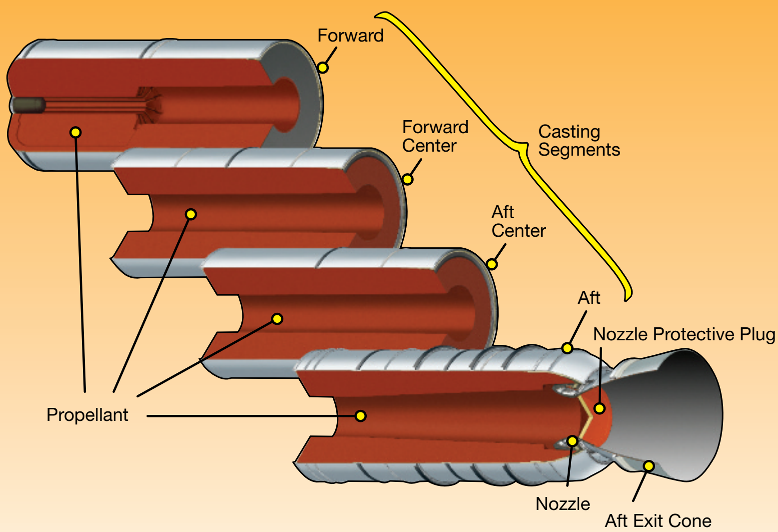
Producing an accurate insulating layer was critical. Too little insulation, and the steel could be heated and melted by the 2,760°C (5,000°F) combustion gases. Too much insulation, and weight requirements were exceeded. Engineers employed sophisticated design analysis and testing to optimize this balance between protection and weight. By design, much of the insulation was burned away during the 2 minutes of motor operation.

The propellant was formulated from three major ingredients: aluminum powder (fuel); ammonium perchlorate (oxidizer); and a synthetic polymer binding agent. The ingredients were batched, fed into large 2,600-L (600-gal) mix bowls, mixed, and tested before being poured into the insulated and lined segments. Forty batches were produced to fill each case segment. The propellant mixture had an initial consistency similar to that of peanut butter, but was cured to a texture and color that resembled a rubber pencil eraser—strong, yet pliable. The propellant configuration or "shape" inside each segment was carefully designed and cast to yield the precise thrust trace upon ignition.

Once each segment was insulated and cast with propellant and finalized, the segments were shipped from ATK's manufacturing facility in Utah to Kennedy Space Center (KSC) in Florida, on specially designed, heavy-duty covered rail cars. At KSC, they were stacked and assembled into the flight configuration.

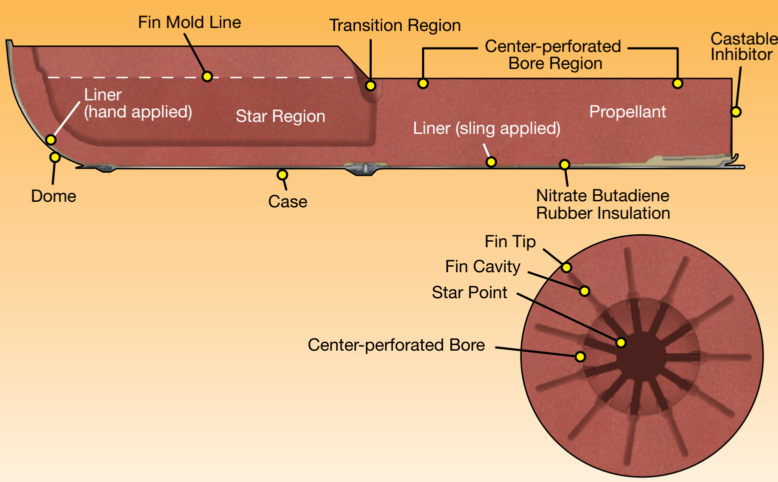
The segments were joined together with tang/clevis joints pinned in 177 locations and sealed with redundant O-rings. Each joint, with its redundant seals and multiple redundant seal protection features, was pressure checked during assembly to ensure a good pressure seal.

Reusable Solid Rocket Motor Propellant Configuration



The four primary propulsion segments that comprised the reusable solid rocket motor were manufactured individually then assembled for launch. Each segment was reusable and designed for a service life of up to 20 flights.

Forward Segment Propellant Grain Configuration



The forward propulsion segment featured a unique grain pattern designed to yield the greatest thrust when it was needed most—on ignition.

An igniter was installed in the forward end of the forward segment—at the top of the rocket. The igniter was essentially a smaller rocket motor that fired into the solid rocket motor to ignite the main propellant grain. Design and manufacture closely mirrored the four main segments.

The nozzle was installed at the aft end of the aft segment, at the bottom of the rocket. The nozzle was the "working" component of the rocket in which hot exhaust gases were accelerated and directed to achieve performance requirements and vehicle control.

The nozzle structure consisted of metal housings over which were bonded layers of carbon/phenolic and silica/phenolic materials that protected the metal structure from the searing exhaust gases by partially decomposing and ablating. A flexible bearing, formed with vulcanized rubber and steel, allowed for nozzle maneuverability up to 8 degrees in any direction to steer the shuttle during the first minutes of flight.

Engineers employed significant analysis and testing to develop a reliable and efficient nozzle capable of being manufactured. The nozzle flexible bearing—measuring up 2.35 m (92.4 in.) at its outside diameter—was an example of one component that required multiple processing iterations to ensure the manufactured product aligned with design requirements.

NASA enhanced the nozzle design following the Challenger accident when severe erosion on one section of the nozzle on one motor was noted through postflight analysis. While the phenolic liners were designed to erode smoothly and predictably, engineers found—at certain ply orientations—that internal stresses resulting from exposure to hot gases exceeded the material strength. Under such stress, the hot charred material had the potential to erode erratically and jeopardize component integrity. Engineers modified nozzle ply angels to reduce material stress, and this condition was successfully eliminated on all subsequent flights.

Technicians shown installing igniter used to initiate the propellant burn in a forward motor segment. The igniter was a small rocket motor loaded with propellant that propagated flame down the bore of the motor.

The Reusable Rocket

All metal hardware—including structures from the case, igniter, safe-and-arm device, and nozzle—were designed to support up to 20 shuttle missions. This was unique to the reusable solid rocket motor. Besides the benefits of conservation and affordability, the ability to recover the motors allowed NASA to understand exactly how the components performed in flight. This performance analysis provided a wealth of valuable information and created a synergy to drive improvements in motor performance, implemented through motor manufacturing and processing.

This recovery and postflight capability was particularly important for the long-term Space Shuttle Program since, over time, changes were inevitable. Change to design or process became mandatory as a result of factors such as material/vendor obsolescence or new environmental regulations.

Changing Processes

During a 10-year period beginning in the mid 1990s, for example, more than 100 supplier materials used to produce the reusable solid rocket motor became obsolete. The largest contributing factor stemmed from supplier economics, captured in three main scenarios. First, suppliers changed their materials or processes. Second, suppliers consolidated operations and either discontinued or otherwise modified their materials. Third, the materials were simply no longer available from subtier vendors.

US environmental regulations, such as the requirement to phase out the use of ozone-depleting chemicals, were an additional factor. Methyl chloroform, for example, was a solvent used extensively in hardware processing. A multimillion-dollar effort was launched within NASA and ATK to eventually eliminate methyl chloroform use altogether in motor processing. Eight alternate materials were selected following thorough testing and analysis to ensure program performance was not compromised.

New Technology

Advancements in technology that occurred during the decades-long program were a further source of change. Engineers incorporated new technologies into motor design and processing as the technology could be proven. Incorporating braided carbon fiber material as a thermal barrier in the nozzle-to-case joint is one example.

Postflight Analysis

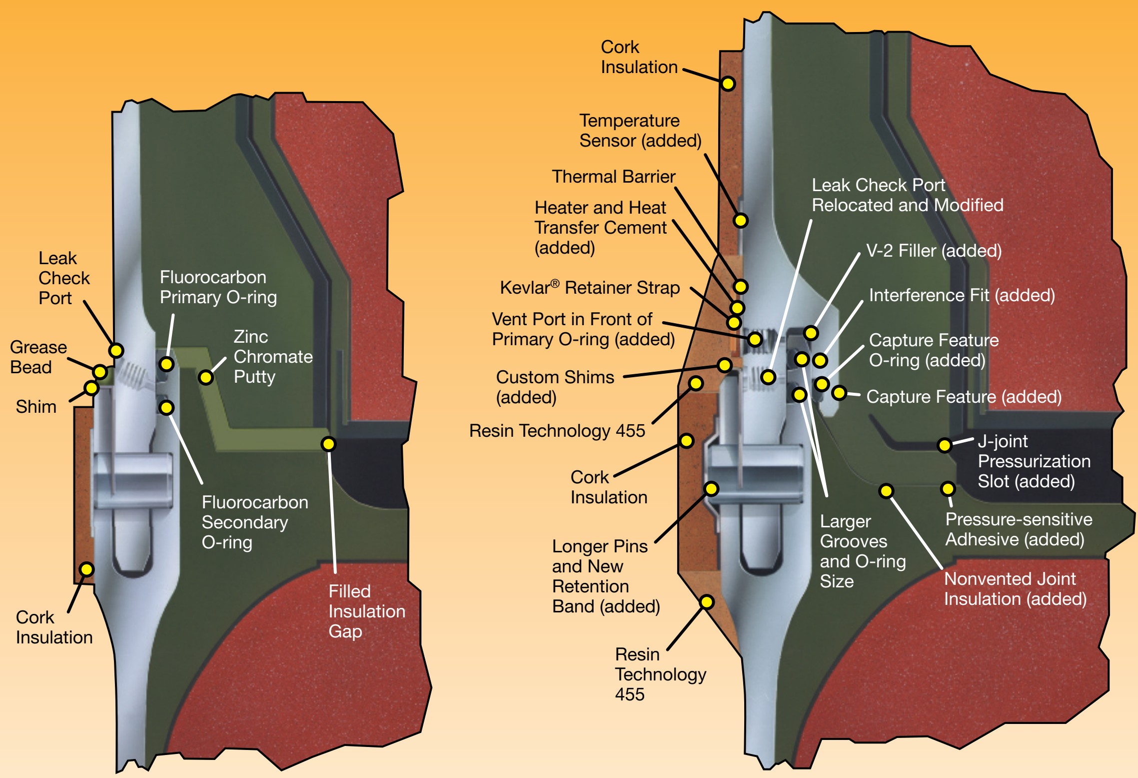
The ability to closely monitor flight performance through hands-on postflight analysis—after myriad material, design, and process changes—was only possible by virtue of the motor's reusable nature.

Developing methods to scrutinize and recertify spent rocket motor hardware that had raced through the stratosphere at supersonic speeds was new. NASA had the additional burden of working with components that had experienced splashdown loads and were subsequently soaked in corrosive saltwater prior to retrieval.

In the early days of the program, NASA made significant efforts in identifying relevant evaluation criteria and establishing hardware assessment methods. A failure to detect hardware stresses and material weaknesses could result in an unforgivable catastrophic event later on. The criteria used to evaluate the first motors and the accompanying data collected would also become the benchmark from which future flights would be measured. Included in the evaluation criteria were signs of case damage or material loss caused by external debris; integrity of major components such as case segments, nozzle and igniter; and fidelity of insulation, seals, and joints.

Inspection and documentation of retrieved hardware occurred in two parts of the country: Florida, where the hardware was retrieved; and Utah, where it underwent in-depth inspection and refurbishment. On recovery, a team of 15 motor engineers conducted what was termed an "open assessment," primarily focusing on exterior components. After retrieval, teams of specialists rigorously dissected, measured, sampled, and assessed joints,

Field Joint Comparison for Use on Reusable Solid Rocket Motor



High-performance Motor | **Reusable Solid Rocket Motor**

Reusable solid rocket motors incorporated significant improvements over the earlier shuttle motors in the design of the joints between the main segments. Redesign of this key feature was part of the intensive engineering redesign and demonstration feat accomplished following the Challenger accident. The result was a fail-safe joint/seal configuration that, with continued refinement, had a high demonstrated reliability. Each joint, with its redundant seals and multiple redundant seal protection features, could be pressure checked during assembly to ensure a good pressure seal was achieved. A similar design approach was implemented on the igniter joints during that same time period.

bondlines, ablatives, fasteners, and virtually all remaining flight hardware. Engineers promptly evaluated any significant observations that could affect the orbiting vehicle or the next motor launch sets.

Before the motor was returned to the flight inventory, the recovered metal parts were inspected for corrosion, deformations, cracks, and other potential damage. Dimensional measurement data were fed into a system-wide database containing documentation dating back to the program's inception. The wealth of information available for performance trend analysis was unmatched by any other solid rocket motor manufacturing process in the world. Gates and checks within the system ensured the full investigation of any anomalies to pinpoint root cause and initiate corrective action.

The postflight analysis program collected the actual flight performance data—most of which would not have been available if the motors had not been recovered.

Through this tightly defined process, engineers were able to address the subtle effects that are often a result of an unintended drift in the manufacturing process or new manufacturing materials introduced into the process. The

In Utah, rigorous test program included 53 reusable solid rocket motor ground tests between 1977 and 2010. Spectators flocked by the thousands to witness firsthand the equivalent of 15 million horsepower safely unleashed from a vantage point of 2 to 3 km (1 to 2 miles) away.

process addressed these concerns in the incipient phase rather than allowing for a potentially serious issue to escalate undetected. The ultimate intangible benefit of this program was greater reliability, as demonstrated by the following two examples.

Postflight assessment of nozzle bondlines was a catalyst to augment adhesive bonding technology and substantially improve hardware quality and reliability. Storage controls for epoxy adhesives were established in-house and with adhesive suppliers. Surface preparation, cleanliness, adhesive primer, and process timelines were established. Adhesive bond quality and robustness were increased by an order of magnitude.

Postflight inspections also occasionally revealed gas paths through the nozzle-to-case joint polysulfide thermal barrier that led to hot gas impingement on the wiper O-ring—a structure protecting the primary O-ring from thermal damage. While this condition did not pose a flight risk, it did indicate performance failed to meet design intent. The root cause: a design that was impossible to manufacture perfectly every time. Engineers resolved this concern by implementing a nozzle-to-case joint J-leg design similar to that successfully used on case field joints and igniters.

Robust Systems Testing

The adage "test before you fly," adopted by the Space Shuttle Program, was the standard for many reusable solid rocket motor processes and material, hardware, and design changes. What ATK, the manufacturer, was able to learn from the vast range of data collected and processed through preflight and ground testing ensured the highest levels of dependability and safety for the hardware. Immediate challenges posed by the 570-metric-ton (1.2-million-pound) motor included handling, tooling, and developing a 17.8-meganewton (4,000,000-pound-force) thrust-capable ground test stand; and designing a 1,000-channel data handling system as well as new support systems, instrumentation capability, data acquisition, and countdown procedures.

Hot-fire testing of full-scale rocket motors in the Utah desert became a hallmark of the reusable solid rocket motor development and sustainment program. Individual motor rockets were fired horizontally, typically once or twice a year, lighting up the mountainside with the brightness of a blazing sun, even in broad daylight.

Following a test firing, quick-look data were available within hours. Full data analyses required several months.

On average, NASA collected between 400 and 700 channels of data for each test. Instrumentation varied according to test requirements but typically included a suite of sensors not limited to accelerometers, pressure transducers, calorimeters, strain gauges, thermocouples, and microphones. Beyond overall system assessment and component qualification, benefits of full-scale testing included the opportunity to enhance engineering expertise and predictive skills, improve engineering techniques, and conduct precise margin testing. The ability to tightly measure margins for many motor process, material, components, and design parameters provided valuable verification data to demonstrate whether even the slightest modification was safe for flight.

Quick-look data revealed basic ballistics performance—pressure and thrust measurements—that could be compared with predicted performance and historic data for an initial assessment.

Full analysis included scrutiny of all data recorded during the actual test as well as additional data gathered from visual inspections and measurements of disassembled hardware, similar to that of postflight inspection. Engineers assessed specific data tied to test objectives. When qualifying a new motor insulation, for example, posttest inspection would additionally include measurements of remaining insulation material to calculate the rate of material loss.

Subscale propellant batch ballistics tests, environmental conditioning testing, vibration tests, and custom sensor development and data acquisition were also successful components of the program to provide specific reliability data.

Culture of Continual Improvement

The drive to achieve 100% mission success, paired with the innovations of pre- and postflight testing that allowed performance to be precisely quantified, resulted in an operating culture in which the bar was continually raised.

Design and processing improvements were identified, pursued, and implemented through the end of the program to incrementally reduce risk and waste. Examples of relatively late program innovations included: permeable carbon fiber rope as a thermal protection element in various nozzle and nozzle/case joints; structurally optimized bolted joints; reduced stress forward-grain fin transition configuration; and improved adhesive bonding systems.

This culture, firmly rooted in the wake of the Challenger accident, led to a comprehensive process control program with systems and tools to ensure processes were appropriately defined, correctly performed, and adequately maintained to guarantee reliable and repeatable product performance.

Noteworthy elements of the motor process control program included an extensive chemical fingerprinting program to analyze and monitor the quality of vendor-supplied materials, the use of statistical process control to better monitor conditions, and the comprehensive use of witness panels—product samples captured from the live manufacturing process and analyzed to validate product quality.

With scrupulous process control, ATK and NASA achieved an even greater level of understanding of the materials and processes involved with reusable solid rocket motor processing. As a result, product output became more consistent over the life of the program. Additionally, partnerships with vendors and suppliers were strengthened as increased performance measurement and data sharing created a win-win situation.

An Enduring Legacy

The reusable solid rocket motor was more than an exceptional rocket that safely carried astronauts and hundreds of metric tons of hardware into orbit for more than 25 years. Throughout the Reusable Solid Rocket Motor Program, engineers and scientists generated the technical know-how in design, test, analysis, production, and process control that is essential to continued space exploration. The legacy of the first human-rated reusable solid rocket motor will carry on in future decades. In the pages of history, the shuttle reusable solid rocket motor will be known as more than a stepping-stone. It will also be regarded as a benchmark by which future solid-propulsion systems will be measured.

Orbital Propulsion Systems— Unique Development Challenges

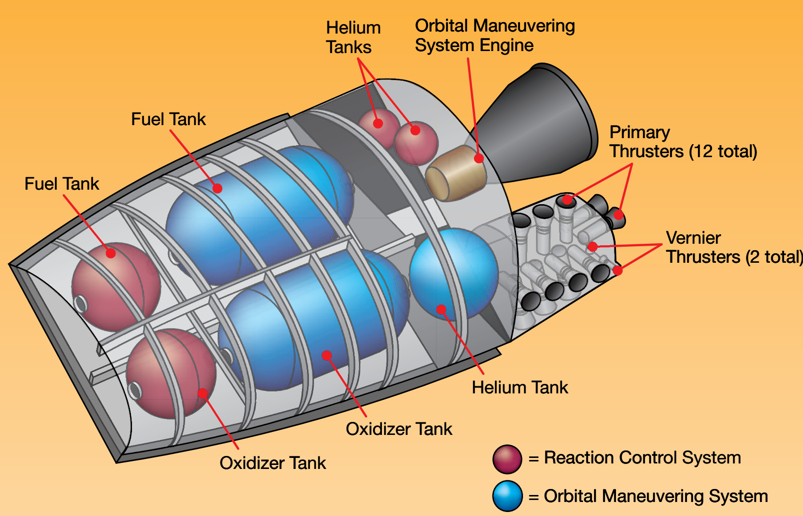
Until the development of the Space Shuttle, all space vehicle propulsion systems were expendable. Influenced by advances in technologies and materials, NASA decided to develop a reusable propulsion system. Although reusability saved overall costs, maintenance and turnaround costs offset some of those benefits.

NASA established a general redundancy requirement of fail operational/fail safe for these critical systems: Orbital Maneuvering System, Reaction Control System, and Auxiliary Power Unit. In addition, engineers designed the propulsion systems for a life of 100 missions or 10 years combined storage and operations. Limited refurbishment was permitted at the expense of higher operational costs.

Orbital Maneuvering System

The Orbital Maneuvering System provided propulsion for the Orbiter during orbit insertion, orbit circularization, orbit transfer, rendezvous, and deorbit. NASA faced a major challenge in selecting the propellant. The agency originally chose liquid oxygen and liquid hydrogen propellants. However, internal volume constraints could not be met for a vehicle configuration that provided a payload of 22,680 kg (50,000 pounds) in a bay measuring 4.6 m (15 ft) in diameter and 18.3 m (60 ft) in length. This, coupled with concerns regarding complexity of cryogenic propellants, led to the consideration of storable hypergolic propellants.

Orbital Maneuvering System/Reaction Control System



Orbital Maneuvering System/Reaction Control System pods viewed from the underside.

NASA ultimately selected monomethylhydrazine as the fuel and nitrogen tetroxide as the oxidizer for this system. As these propellants were hypergolic—they ignited when coming into contact with each other—no ignition device was needed. Both propellants remained liquid at the temperatures normally experienced during a mission. Electrical heaters prevented freezing during long periods in orbit when the system was not in use.

Modular Design Presents Obstacles for Ground Support

Trade studies and design approach investigations identified challenges and solutions. For instance, cost and weight could be reduced with a common integrated structure for the Orbital Maneuvering System and Reaction Control System. This integrated structure was combined with the selection of nitrogen tetroxide and monomethylhydrazine propellants.

Thus, NASA adopted an interconnect system in which the Reaction Control System used Orbital Maneuvering System propellants because of cost, weight, and lower development risk.

Disadvantages of a storable propellant system were higher maintenance requirements resulting from their corrosive nature and hazards to personnel exposed to the toxic propellants. NASA partially addressed these considerations by incorporating the Orbital Maneuvering System into a removable modular pod. This allowed maintenance and refurbishment of those components exposed to hypergols to be separated from other turnaround activities.

For ground operations, it was not practical to remove modules for each turnaround activity, and sophisticated equipment and processes were required for servicing between flights. Fluid and gas connections to the propellants and pressurants used quick disconnects to allow servicing on the launch pad, in Orbiter processing facilities, and in the hypergolic maintenance facility. However, quick disconnects occasionally caused problems, including leakage that damaged Orbiter thermal tiles.

Engineers tested and evaluated many ground support equipment design concepts at the White Sands Test Facility (WSTF). In particular, they tested, designed, and built the equipment used to test and evaluate the propellant acquisition screens inside the propellant tanks before shipment to Kennedy Space Center for use on flight vehicles. The Orbital Maneuvering System/Reaction Control System Fleet Leader Program used existing qualification test articles to detect and evaluate “life-dependent” problems before these problems affected the shuttle fleet. This program provided a test bed for developing and evaluating ground support equipment design changes and improving processes and procedures. An example of this was the Reaction Control System Thruster Purge System, which used low-pressure nitrogen to prevent propellant vapors from accumulating in the thruster chamber. This WSTF-developed ground support system proved beneficial in reducing the number of in-flight thruster failures.

Henry Pohl
Director of Engineering at Johnson Space Center (1986-1993).

"To begin to understand the challenges of operating without gravity, imagine removing the commode from your bathroom floor, bolting it to the ceiling. And then try to use it. You would then have a measure of the challenges facing NASA."

Additional Challenges

Stable combustion was a concern for NASA. In fact, stable combustion has always been the most expensive schedule-constraining development issue in rocket development. For the Orbital Maneuvering System engine, engineers investigated injector pattern designs combined with acoustic cavity concepts. In propulsion applications with requirements for long-duration firings and reusability, cavities had an advantage because they were easy to cool and therefore less subject to failure from either burnout or thermal cycling.

To accomplish precise injector fabrication, engineers implemented platelet configuration. The fuel and oxidizer flowed through the injector and impinged on each other, causing mixing and combustion. Platelet technology, consisting of a series of thin plates manufactured by photo etching and diffusion bonded together, eliminated mechanical manufacturing errors and increased injector life and combustion efficiency.

The combustion chamber was regenerative-cooled by fuel flowing in a single pass through non-tubular coolant channels. The chamber was composed of a stainless-steel liner, an electroformed nickel shell, and an aft flange and fuel inlet manifold assembly. Its structural design was based on life

Formation of Metal Nitrates Caused Valve Leaks

Being the first reusable spacecraft—and in particular, the first to use hypergolic propellants—the shuttle presented technical challenges, including leaky and sticky propellant valves in the Reaction Control System thrusters. Early in the program, failures in this system were either an oxidizer valve leak or failure to reach full chamber pressure within an acceptable amount of time after the thruster was commanded on. NASA attributed both problems to the buildup of metal nitrates on and around the valve-sealing surfaces.

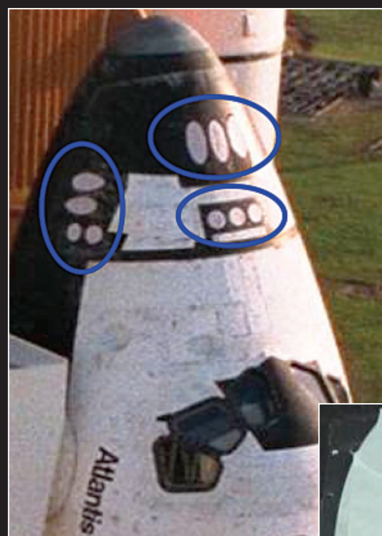
Metal nitrates were products of iron dissolved in the oxidizer when purchased and iron and nickel that were leached out of the ground and flight fluid systems. When the oxidizer was exposed to reduced pressure or allowed to evaporate, metal nitrates precipitated out of solution and contaminated the valve seat.

Subsequent valve cycling caused damage to the Teflon® valve seat, further exacerbating the leakage until sufficient nitrate deposition resulted in "gumming" up the valve. At that point, the valve was either slow to operate or failed to operate.

Multiple changes reduced the metal nitrate problem but may have contributed to fuel valve seat extrusion, which manifested years later. The fuel valve extrusion was largely attributed to the use of throat plugs. These plugs trapped oxidizer vapor leakage in the combustion chamber, which subsequently reacted at a low level of fuel that had permeated the Teflon® fuel valve seat. This problem was successfully addressed with the implementation of the NASA-developed thruster nitrogen purge system, which kept the thruster combustion chamber relatively free of propellant vapors.

An Ordinary Solution to the Extraordinary Challenge of Rain Protection

During operations, Orbiter engines needed rain protection after the protective structure was moved away and protective ground covers were removed. This requirement protected the three upward-facing engines and eight of the left-side engines from rainwater accumulation on the launch pad. The up-firing engine covers had to prevent water accumulation that could freeze in the injector passages during ascent. The side-firing engine covers prevented water from accumulating in the bottom of the chamber and protected the chamber pressure sensing ports. Freezing of accumulated water during ascent could block the sensing port and cause the engine to be declared "failed off" when first used. The original design concept allowed for Teflon® plugs installed in the engine throats and a combination of Teflon® plugs tied to a Teflon® plate that covered the nozzle exit. This concept added vehicle weight, required special procedures to eject the plugs in flight, and risked accidental ejection in ascent that could damage tiles. The solution used ordinary plastic-coated freezer paper cut to fit the exit plane of the nozzle. Tests proved this concept could provide a reliable seal under all expected rain and wind conditions. The covers were low cost, simple, and added no significant weight. The thruster rain cover material was changed to Tyvek® when NASA discovered pieces of liberated plastic-coated paper beneath the cockpit window pressure seals. The new Tyvek® covers were designed to release at relatively low vehicle velocity so that the liberated covers did not cause impact damage to windows, tile, or any other Orbiter surface.



Tyvek® covers shown installed on forward Reaction Control System thrusters (top) and a typical cover (right). Note that the covers were designed to fit certain thruster exit plane configurations.

cycle requirements, mechanical loads, thrust and aerodynamic loading on the nozzle, ease of fabrication, and weight requirements.

The nozzle extension was radiation cooled and constructed of columbium metal consistent with experience gained during the Apollo Program. The mounting flange consisted of a bolt ring, made from a forging and a tapered section, that could either be spun or made from a forging. The forward and aft sections were made from two panels each. This assembly was bulge formed to the final configuration and the stiffening rings were attached by welding. The oxidation barrier diffusion operation was done after machining was completed.

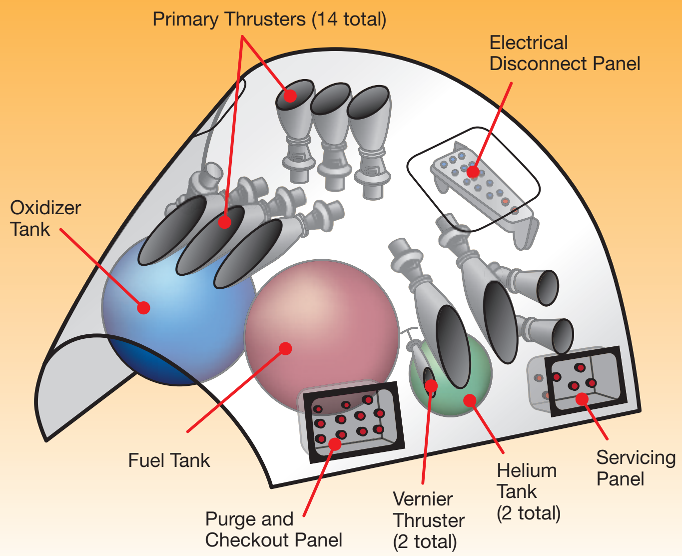
A basic design challenge for the bipropellant valve was the modular valve. The primary aspect of the assembly design was modularization, which reduced fabrication problems and development time and allowed servicing and maintenance goals to be met with lower inventory.

NASA Seeks Options as Costs Increase

The most significant lesson learned during Orbital Maneuvering System development was the advantage of developing critical technologies before initiating full-scale hardware designs. The successful completion of predevelopment studies not only reduced total costs, also it minimized schedule delays.

In the 1980s, NASA began looking for ways to decrease the cost of component refurbishment and repair. NASA consolidated engineering, evaluation, and repair capabilities for many components, and reduced overall costs. Technicians serviced, acceptance

Forward Reaction Control System



Forward Reaction Control System on Discovery.

tested, and prepared all hypergolic wetted components for reinstallation on the vehicles.

Reaction Control System

The Reaction Control System provided propulsive forces to control the motion of the Orbiter for attitude control, rotational maneuvers, and small velocity changes along the Orbiter axes. The requirement of a fail-operational/fail-safe design introduced complexity of additional hardware and a complex critical redundancy management system. The reuse requirement posed problems in material selection and compatibility, ground handling and turnaround procedures, and classical wear-out problems. The requirement for both on-orbit operations and re-entry into Earth's atmosphere complicated propellant tank acquisition system design because of changes in the gravitational environment.

NASA Makes Effective Selections

As with the Orbital Maneuvering System, propellant selection was important for the Reaction Control System. NASA chose a bipropellant of monomethylhydrazine and nitrogen

Low Temperatures, Increased Leakage, and a Calculated Solution

Some primary thruster valves could leak when subjected to low temperature. NASA discovered this problem when they observed liquid dripping from the system level engines during a cold environment test. The leakage became progressively worse with increased cycling. Continued investigation indicated that tetrafluoroethylene Teflon® underwent a marked change in the thermal expansion rate in a designated temperature range. Because machining, done as a part of seat fabrication, was accomplished in this temperature range, some parts had insufficient seat material exposed at reduced temperatures.

To reduce susceptibility to cold leakage, engineers machined Teflon® at 0°C (32°F) to ensure uniform dimensions with adequate seat material exposed at reduced temperatures and raised the thruster heater set points to maintain valve temperature above 16°C (60°F).

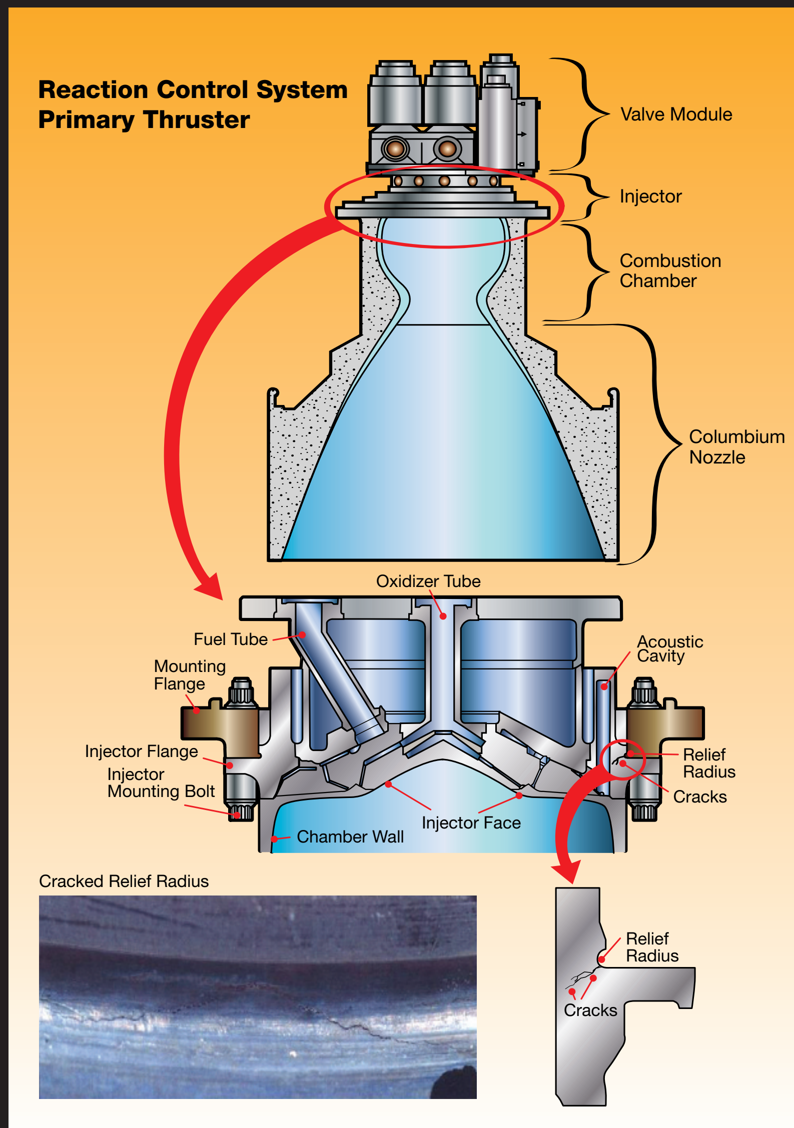
Cracks Prompt Ultrasonic Inspection

Late in the Space Shuttle Program, NASA discovered cracks in a thruster injector. The thruster was being refurbished at White Sands Test Facility (WSTF) during the post-Columbia accident Return to Flight time period. The cracks were markedly similar to those that had occurred in injectors in 1979 and again in 1982.

These earlier cracks were discovered during manufacturing of the thrusters and occurred during the nozzle insulation bake-out process. Results from the laboratory testing indicated that cracks were developed due to chemical processing and manufacturing.

In addition to using leak testing to screen for injector cracking, NASA engineers developed and implemented an ultrasonic inspection procedure to screen for cracks that measured less than the injector wall thickness.

The marked similarity of the crack location and crack surface appearance strongly suggested the WSTF-discovered cracks were due to the original equipment manufacturing process and were not flight induced or propagated. Laboratory tests and analyses confirmed that those cracks were induced in manufacturing. The cracks had not grown significantly over the years of the thruster's use and its many engine firings. Laboratory nondestructive testing showed that the original ultrasonic inspection process was not very reliable and it was possible that manufacturing-induced cracks could escape detection and cracked thrusters could have been placed in service. The fact that there was no evidence of crack growth associated with the WSTF-discovered cracks due to the service environment was a significant factor in the development of flight rationale for the thrusters.



Reaction Control System thruster cross sections showing the crack location and its actual surface appearance.

tetroxide system, which allowed for integration of this system with the Orbital Maneuvering System. This propellant combination offered a favorable weight tradeoff, reasonable development cost, and minimal development risk.

NASA selected a screen tank as a reusable propellant supply system to provide gas-free propellants to the thrusters. Screen tanks worked by using the surface tension of the liquid to form a barrier to the pressurant gas. The propellant acquisition device was made of channels covered with a finely woven steel mesh screen. Contact with liquid wetted the screen and surface tension of the liquid prevented the passage of gas. The strength of the liquid barrier was finite. The pressure differential at which gas would be forced through the wetted screen was called the "bubble point." When the bubble point was exceeded, the screen broke down and gas was transferred. If the pressure differential was less than the bubble point, gas could not penetrate the liquid barrier and only liquid was pulled through the channels. NASA achieved their goal in designing the tank to minimize the pressure loss while maximizing the amount of propellant expelled.

Several Reaction Control System component failures were related to nitrate contamination. Storage of oxidizer in tanks and plumbing that contained iron caused contamination in the propellant. This contamination formed a nitrate that could cause valve leakage, filter blockage, and interference in sliding fits. The most prominent incident was the failure of a ground half-quick disconnect to close, resulting in an oxidizer spill on the launch pad. NASA implemented a program to determine the parameters that caused the iron nitrate formation and implement procedures to prevent its formation in the future. This resulted in understanding the relationship between iron, water, nitric oxide content, and nitrate formation. The agency developed production and storage controls as well as filtration techniques to remove the iron, which resolved the iron nitrate problem.

Auxiliary Power Unit

The Auxiliary Power Unit generated power to drive hydraulic pumps that produced pressure for actuators to control the main engines, aero surfaces, landing gear, brakes, and nose wheel steering. The Auxiliary Power Unit shared common hardware and systems with the Hydraulic Power Unit used on the solid rocket motors. The shuttle needed a hydraulic power unit that could operate from zero to three times gravity, at vacuum and sea-level pressures, from -54°C to 107°C (-65°F to 225°F), and be capable of restarting. NASA took the basic approach of using a small, high-speed, monopropellant-fuel, turbine-powered unit to drive a conventional aircraft-type hydraulic pump.

If the Auxiliary Power Unit was restarted before the injector cooled to less than 204°C to 232°C (400°F to 450°F), the fuel would thermally decompose behind the injector panels and damage the injector and the Gas Generator Valve Module. Limited hot-restart capability was achieved by adding an active water cooling system to the gas generator to be used only for hot restarts. This system injected water into a cavity within the injector. The steam generated was vented overboard. Use of this system enabled restarts at any time after the cooling process, which required a 210-second delay.

Improved Machining and Manufacturing Solves Valve Issue

Development of a reliable valve to control fuel flow into the gas generator proved to be one of the most daunting tasks of the propulsion systems. The valve was required to pulse fuel into the gas generator at frequencies of 1 to 3 hertz. Problems with the valve centered on leakage and limited life due to wear and breakage of the tungsten carbide seat. NASA's considerable effort in redesigning the seat and developing manufacturing processes resulted in an intricate seat design with concentric dual sealing surfaces and redesigned internal flow passages. The seat was diamond-slurry honed as part of the manufacturing process to remove the recast layer left by the electro-discharge machining. This recast layer was a source of stress risers and was considered one of the primary factors causing seat failure. The improved design and machining and manufacturing processes were successful.

Additional Challenges and Subsequent Solutions

During development testing of the gear box, engineers determined that the oil pump may not funtion satisfactorily on orbit due to low pressure. It became necessary to provide a fluid for the pump to displace to assure the presence of oil at the inlet and to have a mechanism to provide needed minimum pressure at startup and during operation.

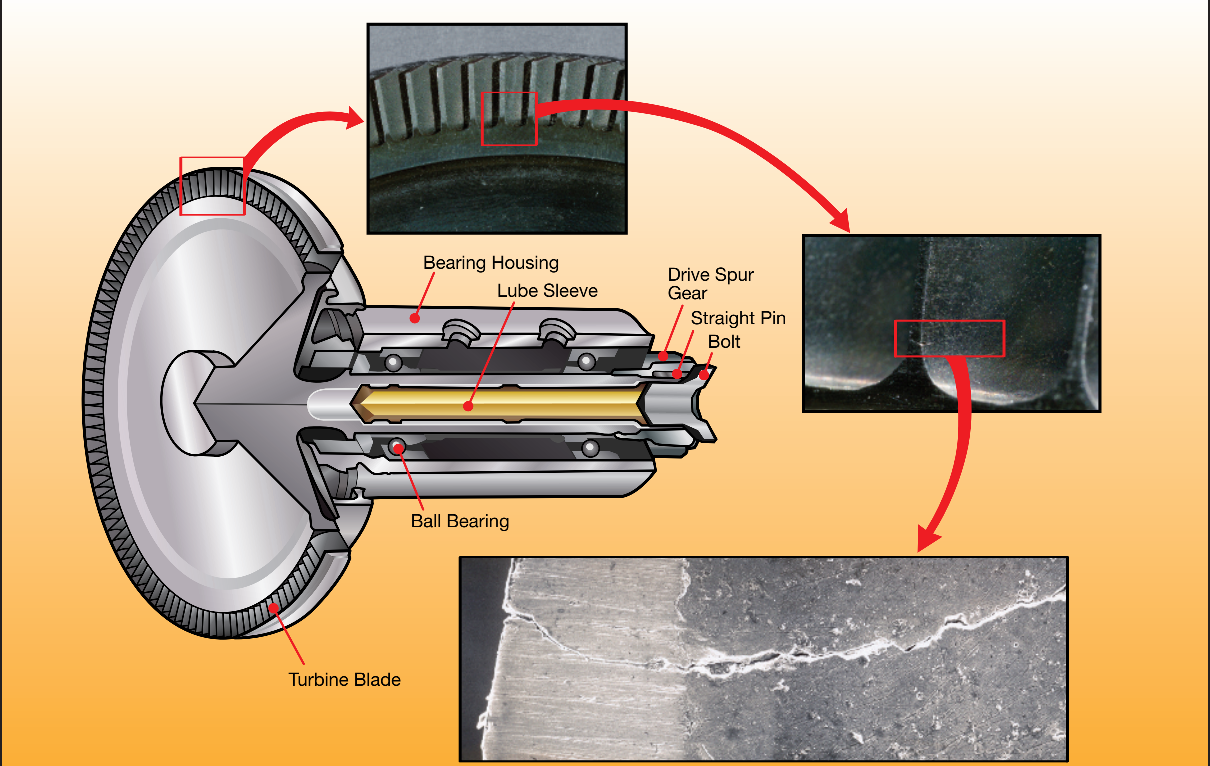
The Auxiliary Power Unit was designed with a turbine wheel radial containment ring and a blade tip seal and rub ring to safely control failures of the high-speed assembly. The containment ring was intended to keep any wheel fragments from leaving the Auxiliary Power Unit envelope. NASA provided safety features that would allow operation within the existing degree of containment. The agency used an over-speed safety circuit to automatically shut down a unit at 93,000 revolutions per minute. To provide further insurance against wheel failure, NASA imposed stringent flaw detection inspections. With these controls, results of fracture mechanics analyses showed the theoretical life to be 10 times the 100-mission requirement.

With these improvements, the Auxiliary Power Unit demonstrated success of design and exhibited proven durability, performance, and reusability.

NASA Encounters Obstacle Course in Turbine Wheel Design

The space agency faced multiple challenges with the development of the turbine wheel. Aerodynamically induced high-cycle fatigue caused cracking. Analysis indicated this part of the blade could be removed with a small chamfer at the blade tip without significant effect on performance. This cracking problem was resolved by careful design and control of electromechanical machining.

The shroud cracking problem was related to material selection and the welding process. Increased strength and weld characteristics were achieved by changing the shroud material. Engineers developed a controlled electron beam weld procedure to ensure no overheating of the shroud. These actions eliminated the cracking problem.

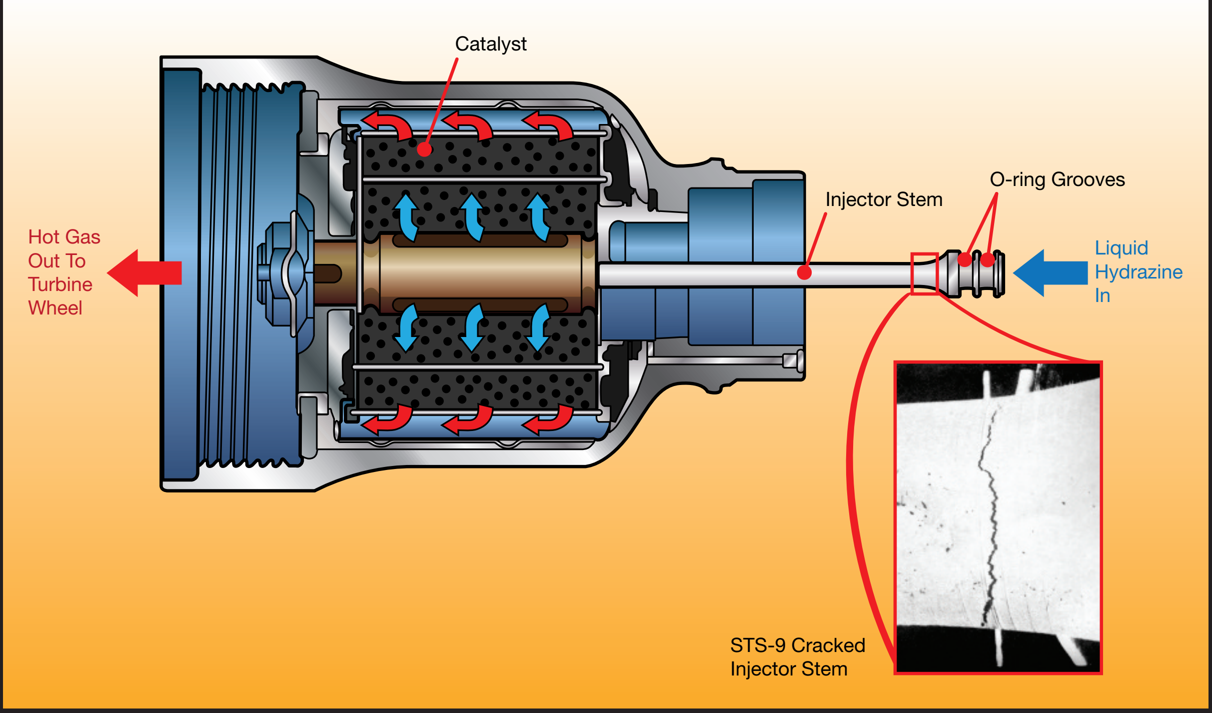


Blade Root Cracking

Stress Corrosion and Propellant Ignition

One of the most significant Auxiliary Power Unit problems occurred during the STS-9 (1983) mission when two of the three units caught fire and detonated. Postflight analysis indicated the presence of hydrazine leaks in Auxiliary Power Units 1 and 2 when they were started for re-entry while still in orbit. The leaking hydrazine subsequently ignited and the resulting fire overheated the units, causing the residual hydrazine to detonate after landing. The fire investigation determined the source of the leaks to be nearly identical cracks in the gas generator injector tubes in both units. Laboratory tests further determined that the injector tube cracks were due to stress corrosion from ammonium hydroxide vapors generated by decomposition of hydrazine in the catalyst bed after Auxiliary Power Unit shutdown.

Initial corrective actions included removal of the electrical machined recast layer on the tube inside diameter and an improved assembly of the injector tube. Later, resistance to stress corrosion and general corrosion was further improved by chromizing the injector tubes.



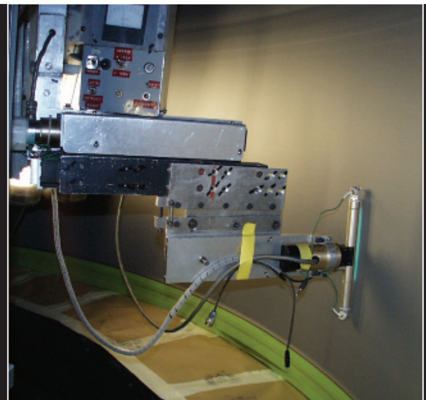
Summary

The evolution of orbital propulsion systems for the Space Shuttle Program began with Apollo Program concepts, expanded with new technologies required to meet changing requirements, and continued with improvements based on flight experience. The design requirements for 100 missions, 10 years, and reuse presented challenges not previously encountered. In addition, several problems were not anticipated. NASA met these challenges, as demonstrated by the success of these systems.

Pioneering Inspection Tool

Contamination Scanning of Bond Surfaces

Bonding thermal insulation to metal case surfaces was a critical process in solid rocket motor manufacturing during the Space Shuttle Program. Surfaces had to be immaculately clean for proper adherence. The steel alloy was susceptible to corrosion and was coated with grease for protection during storage. That grease, and the solvents to remove it, became potential contaminants.



Inspection technology capitalizing on the photoelectric effect provided significant benefits over the traditional method of visual inspection using handheld black lights. The technology was developed through a NASA/industry partnership managed by Marshall Space Flight Center. Specific benefits included increased accuracy in contamination detection and an electronic data record for each hardware inspection.

The improvement of contamination inspection techniques was initiated in the late 1980s. The development of a quantitative and recordable inspection technique was based on the physics of optically stimulated electron emission (photoelectric effect) technology being developed at NASA's Marshall Space Flight Center at the time.

Fundamentally, incident ultraviolet light excites and frees electrons from the metal surface. The freed electrons having a negative charge are attracted to a positively charged collector ring in the "Con Scan" (short for Contamination Scanning) sensor. When contamination exists on a metal surface, the amount of ultraviolet radiation that reaches the surface is reduced. In turn, the current is reduced, confirming the presence of a contaminant.

Approximately 90% of each reusable solid rocket motor barrel assembly was inspected using automated Con Scan before bond operations. Technicians mounted the sensor on a robotic arm, which allowed longitudinal translation of the sensor as the barrel assembly rotated on a turntable. Inspection results were mapped, showing color-coded contamination levels (measured current) vs. axial and circumferential locations on the case inner diameter. Color coding made acceptable and rejected areas visually apparent.

By pioneering optically stimulated electron emission technology, which was engineered into a baseline inspection tool, the Space Shuttle Program significantly improved contamination control methods for critical bonding applications.

Propulsion Systems and Hazardous Gas Detection

Shuttle propulsion had hazardous gases requiring development of detection systems including purged compartments. This development was based on lessons learned from the system first used during Saturn I launches.

NASA performed an exhaustive review of all available online monitoring mass spectrometry technology for the shuttle. The system the agency selected for the prototype Hazardous Gas Detection System had an automated high-vacuum system, a built-in computer control interface, and the ability to meet all program-anticipated detection limit requirements.

The instrument arrived at Kennedy Space Center (KSC) in December 1975 and was integrated into the sample delivery subsystem, the control and data subsystem, and the remote control subsystem designed by KSC. Engineers extensively tested the unit for functionality, detection limits and dynamic range, long-term drift, and other typical instrumental performance characteristics. In May 1977, KSC shipped the prototype Hazardous Gas Detection System to Stennis Space Center to support the shuttle main propulsion test article engine test firings. The system remained in use at Stennis Space Center for 12 years and supported the testing of upgraded engines.

The first operational Hazardous Gas Detection System was installed for the system on the Mobile Launch

Platform-1 during the late summer of 1979. Checkout and operations procedure development and activation required almost 1 year, but the system was ready to support initial purge activation and propellant loading tests in late 1980. A special test in which engineers introduced simulated leaks of hydrogen and oxygen into the Orbiter payload bay, lower midbody, aft fuselage, and the External Tank intertank area represented a significant milestone. The system accurately detected and measured gas leaks.

After the new system's activation issues were worked out, it could detect and measure small leaks from the Main Propulsion System. The Hazardous Gas Detection System did not become visible until Space Transportation System (STS)-6—the first launch of the new Orbiter Challenger—during a flight readiness test. In this test, the countdown would proceed normally to launch time, the Orbiter main engines would ignite, but the Solid Rocket Booster engines would not ignite and the shuttle would remain bolted to the launch pad during a 20-second firing of the main engines. The STS-1 firing test for Columbia had proceeded normally, but during Challenger's firing test, the Hazardous Gas Detection System detected a leak exceeding 4,000 parts per million. Rerunning the firing test and performing further leak hunting and analysis revealed a number of faults in the main engines. The manager for shuttle operation propulsion stated that all the money spent on the Hazardous Gas Detection System, and all that would ever be spent, was paid for in those 20 seconds when the leak was detected.

Originally, NASA declined to provide redundancy for the Hazardous Gas Detection System due to a lack of a launch-on-time requirement; however, the agency subsequently decided that redundancy was required. After a detailed engineering analysis followed by lab testing of candidate mass spectrometers, the space agency selected the PerkinElmer MGA-1200 as the basis of the backup Hazardous Gas Detection System. This backup was an ion-pumped, magnetic-sector, multiple-collector mass spectrometer widely used in operating rooms and industrial plants. Although the first systems were delivered in late 1985, full installation on all mobile launch platforms did not occur until NASA completed the Return to Flight activities following the Challenger accident in 1986.

In May 1990, the Hazardous Gas Detection System gained attention once again when NASA detected a hydrogen leak in the Orbiter aft fuselage on STS-35. The space agency also detected a hydrogen leak at the External Tank to Orbiter hydrogen umbilical disconnect and thought that the aft fuselage leakage indication was due to hydrogen from the external leak migrating inside the Orbiter. Workers rolled STS-35 back into the Vertical Assembly Building and replaced the umbilical disconnect. Meanwhile, STS-38 had been rolled to the pad and leakage was again detected at the umbilical disconnect, but not in the aft fuselage. STS-38 was also rolled back, and its umbilical disconnect was replaced. The ensuing investigation revealed that manufacturing defects in both units caused the leaks, but not before STS-35 was back on the pad.

During launch countdown, NASA detected the aft fuselage hydrogen leak. It was then apparent that STS-35 had experienced two separate leaks. The Space Shuttle Program director appointed a special tiger team to investigate the leak problem. This team suspected that the Hazardous Gas Detection System was giving erroneous data, and brought 10 experts from Marshall Space Flight Center to assess the system design. KSC design engineering provided an in-depth, 2-week description of the design and performance details of both the Hazardous Gas Detection System and the backup system. The most compelling evidence of the validity of the readings was that both systems, which used different technology, had measured identical data, and both systems had recorded accurate calibration data before and after leakage detection. After a series of mini-tanking tests—each with increased temporary instrumentation—engineers located and repaired the leak, and STS-35 lifted off for a successful mission on December 9, 1990.

The Hazardous Gas Detection System and backup Hazardous Gas Detection System continued to serve the shuttle until 2001, when both systems were replaced with Hazardous Gas Detection System 2000—a modern state-of-the-art system with a common sampling system and identical twin quadrupole mass spectrometers from Stanford Research Institute. The Hazardous Gas Detection System served for 22 years and the backup Hazardous Gas Detection System served for 15 years.